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RESEARCH MEMORANDUM

GENERAL CONSIDERATIONS OF MACH NUMBER EFFECTS
ON COMPRESSOR-BLADE DESIGN

By John F. Klapproth

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON

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SUMMARY

The combination of high compressor-stage pressure ratio and high mass-flow capacity require the use of Mach numbers relative to the rotor blading greater than 1.0. By operating in the high Mach number range and eliminating the need for high turning inlet-guide vanes, higher pressure ratio stages can be designed without excessive tip loading.

A qualitative picture of the effect of Mach number on blade shape for a prescribed velocity distribution was obtained by the use of an approximate design method. The principal effects of Mach number were to displace the position of maximum thickness toward the rear of the blade and to reduce the maximum thickness. The blade leading-edge radius was small for all examples. The blade shape obtained for a relative inlet Mach number of 1.2 could be closely approximated by a double circular arc.

Experimental performance of rotors having blades similar to those obtained using the approximate design procedure shows rotor efficiencies above 90 percent for rotor tip Mach numbers up to 1.4. No sacrifice in low-speed range or performance was observed by use of blading with small leading-edge radii.

INTRODUCTION

The problems encountered in efforts to obtain supersonic flight speeds have emphasized the need for small engine size and weight, and if possible improved component efficiency. The discussion presented herein is concerned with the possibility of improving the compressor component from aerodynamic considerations and thus to permit a closer realization of the desired over-all engine characteristics. The approach to the problem is based on a reduction in compressor size; it is presumed that, if the size is reduced, the weight can also be reduced and that compressor efficiency can at the same time be maintained or possibly improved.

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The reduction in the size of the compressor should be obtained through reduction of the diameter as well as the length. Reducing the diameter of a compressor requires an increase in the air-flow handling capacity, frequently expressed in pounds per second per square foot of compressor frontal area. Reductions in the length can be obtained by using fewer stages, a larger stage pressure ratio, and approximately the present axial length of each stage.

Factors affecting the flow capacity and the stage pressure ratio of the compressor are discussed and the effects of operation at supersonic relative Mach numbers on stage design characteristics and efficiency are considered in the present paper. A very approximate procedure for designing blades to operate at supersonic relative velocities and a limited amount of experimental-performance data for single-stage rotors designed to operate at supersonic relative Mach numbers are presented.

This work was conducted at the NACA Lewis laboratory.

GENERAL CONSIDERATIONS

Weight-flow capacity. - The weight-flow capacity of a compressor depends primarily on the axial Mach number entering the first rotor-blade row and the annular flow area available. The relation of these factors to flow capacity is shown in figure 1, where weight flow per unit frontal area is plotted against inlet axial Mach number for various rotor hub-tip radius ratios. Standard stagnation conditions ahead of the compressor and the condition of no inlet whirl ahead of the rotor were assumed. The minimum hub-tip radius ratio depends to a large extent on mechanical considerations of stress and vibrations. Within a given hub-tip radius ratio, the flow capacity can only be increased by use of inlet axial Mach numbers higher than the current average of approximately 0.5.

Stage pressure ratio. - The stage pressure ratio is dependent upon the rotor speed and the change in moment of momentum according to

$$\frac{P_2}{P_1} = \left[1 + (\gamma - 1)\ \eta_{ad}\ M_T(R_2 V_{\theta,2} - R_1 V_{\theta,1})\right]^{\frac{\gamma}{\gamma-1}} \qquad (1)$$

and for no radius change through the rotor or for no guide vanes,

$$\frac{P_2}{P_1} = \left[1 + (\gamma - 1)\ \eta_{ad}\ M_T R \Delta V_\theta\right]^{\frac{\gamma}{\gamma-1}}$$

The permissible values of the change in tangential velocity ΔV_θ that can be obtained at good efficiency have been found to be influenced appreciably by the amount of diffusion across the blade row (ref. 1). In order to avoid large losses, the diffusion factor

$$D = (1 - V_2'/V_1') + \Delta V_\theta/2\sigma V_1' \sim \frac{V'_{max} - V_2'}{V_1'} \quad \text{(ref. 1)} \qquad (2)$$

should be held less than approximately 0.45 for the tip regions of an inlet stage.

Figure 2 illustrates the total-pressure ratio attainable with various values of rotor speed and diffusion factor for a constant axial velocity across the blade row and an adiabatic efficiency of 90 percent. Obviously, both a maximum diffusion factor consistant with good efficiency and a high rotor speed are necessary to obtain high stage pressure ratios.

Rotor-inlet Mach number. - The considerations of flow capacity and stage pressure ratio point toward the desirability of using a high inlet axial Mach number and a high rotor speed. However, under these conditions, Mach numbers relative to the rotor blades become appreciably greater than those used in current designs; this is shown in figure 3 where the relative Mach number is plotted as a function of rotor speed and inlet axial Mach number for no inlet guide vanes. The relative Mach numbers are supersonic for all cases except for the low rotational speeds and low axial velocities.

Early attempts to combine the high rotor speeds and high inlet axial velocities usually led to severe sacrifices in rotor efficiency. The rotor designs generally were based on a symmetrical velocity diagram and incorporated an appreciable amount of prewhirl or rotation of the air ahead of the rotor. The performance of such a unit is shown in figure 4 where the peak efficiency at any Mach number is plotted against Mach number (ref. 2). Tests of subsonic airfoils in cascade also showed a marked drag rise with increases in Mach number above about 0.80 (fig. 10(b) of ref. 3). As a result, a Mach number limit was imposed on compressor design which required the use of high guide-vane turning. This Mach number limit restricted the amount of mass flow and the pressure ratio that could be obtained. The influence of a Mach number limit on the design of a typical inlet stage having a 0.5 inlet radius ratio is illustrated in figure 5. Not only is the compromise between mass flow and pressure ratio required, but the generally low value of each is apparent. Consequently, if both high flow capacity and high stage pressure ratio are to be obtained, the designer must use supersonic velocities relative to the rotor blade rows.

Rotor discharge conditions. - The efficiency of the compressor rotor was generally believed to depend primarily upon the inlet relative Mach number. Recent studies, however, (ref. 1) have shown the blade loading (expressed as the diffusion factor in eq. (2)) to be a critical determinant of rotor efficiency. The diffusion factor is influenced not only by the ΔV_θ of the blade row but also by the relative velocity ratio across the blade row; as a result, the flow conditions at the rotor exit in addition to the inlet Mach number can seriously affect the rotor performance.

The discharge conditions at the rotor exit can generally be approximated by using the condition of simple radial equilibrium (that is, $dp/dR = \gamma\rho\ V_\theta^2/R$). If a rotation such as that obtained for the symmetrical velocity diagram at all radii is imparted ahead of the rotor, then a radial variation in the axial velocity at the rotor discharge occurs and the axial velocity is reduced toward the tip. For the case of large amounts of guide-vane turning, such as would occur in any attempt to obtain high rotor speeds and high flow capacity, the reduction in the axial velocity across the rotor tip can be very severe, and the diffusion across the blade row, expressed as $(V_1'-V_2')/V_1'$ (from eq. (2)), may become quite large. The resultant effect is either to restrict the change in tangential velocity and, consequently, the pressure ratio that can be obtained without exceeding diffusion-factor limits for good efficiency, or, for a given pressure ratio, to increase the diffusion factor at the rotor tip well above the level of good efficiency.

If the restrictions on inlet Mach number are removed so that the inlet guide vanes may be eliminated, the radial distribution of axial velocity behind the rotor is constant for the usual case of constant energy addition in the rotor. If a constant average axial velocity across the rotor is prescribed, the tip diffusion for the case of no guide vanes is considerably less than that resulting from designs based on symmetrical velocity diagram having the same weight flow and pressure ratio. If approximately the same limiting value of the diffusion factor is assumed to hold either with or without guide vanes, then the allowable ΔV_θ and the resultant pressure ratio at the rotor tip without guide vanes are appreciably greater than with the guide vanes in designs having the same average axial velocity ratio across the rotor. For example, a comparison of a 0.5 hub-tip radius-ratio rotor of symmetrical diagrams at all radii with a free-vortex transonic design without inlet guide vanes indicates approximately twice the permissible work input for the transonic rotor at the same rotor tip diffusion factor of 0.4.

Efficiency. - The observed efficiency of compressors having inlet guide vanes generally decreases rapidly as the rotor speed is increased above design. This decay in performance is influenced both by the Mach number effects on the flow about the rotor blading and excessive diffusion at the rotor tip. The losses associated with supersonic flow about the rotor blades can be expected to increase with Mach number.

However, if the additional work input obtained by eliminating the guide vanes and using transonic relative Mach numbers increases in proportion to the increased blade losses, then the rotor efficiency can remain constant in spite of the higher blade losses due to Mach number effects (ref. 4). The comparison of the 0.5 hub-tip radius-ratio rotor having a symmetrical diagram at all radii with the free-vortex transonic design indicates that the transonic rotor may have losses twice as great and yet give the same efficiency. If, on the other hand, the allowable loss to give equal efficiency is converted to a loss coefficient by dividing the loss in relative total pressure by the difference between the inlet relative total pressure and the inlet static pressure (ref. 1), then the resultant loss coefficients are approximately constant because of the increased Mach number for the transonic rotor. Thus, if the free-vortex transonic designs are to be utilized to improve stage pressure ratio and flow capacity without sacrifice in efficiency, Mach number effects must be minimized by use of proper blading and careful control of the diffusion factor.

HIGH-SPEED FLOW OVER CONVENTIONAL BLADING

The high-speed flow over conventional blading might be used to furnish an insight into the sources of the increased losses with increased Mach number. Schlieren photographs obtained at the NACA Langley laboratory of a cascade at high Mach numbers are shown in figure 6. A pronounced supersonic expansion is observed near the leading edge of the blade. The supersonic flow must then be decelerated to the subsonic exit Mach number which for the cascade is less than the entering Mach number. This deceleration generally takes the form of a normal shock with an attendant sudden increase in pressure across the shock. For entrance Mach numbers producing only limited regions of slightly supersonic flow, the occurrence of the normal shock does not usually produce a large loss in lift or an increase in drag. However, if the maximum surface Mach number exceeds certain limiting values depending on the Reynolds number and the boundary layer, the normal shock generally is accompanied by a pronounced separation (ref. 5) with a resultant loss in lift and a very large increase in drag. The tendency to form a shock is observed in figure 6 at an entrance Mach number of 0.75. Very pronounced shock formations occur at a Mach number of 0.82, although measured flow separation is not severe. At a Mach number of 0.88, a very definite flow separation occurs, with the separation increasing with Mach number. These observations are similar to those reported in reference 6.

The observed velocity distribution about an airfoil in cascade is shown in figure 7, which illustrates the rapid rise to supersonic velocities near the leading edge, the rapid deceleration associated with the shock pattern, and the further deceleration to the exit velocity. The ΔV_θ produced by the blade will depend upon the area between the curves of suction- and pressure-surface velocities. An equivalent ΔV_θ obtained by a rescheduling of the suction-surface velocities to a more nearly constant value would result in marked reduction in the level of the maximum blade-surface velocity.

The observation that cascades as well as isolated airfoils may operate satisfactorily with local regions of supersonic flow indicates that supersonic Mach numbers in themselves do not constitute an invariable limit. The large static-pressure rise associated with the normal shock may reasonably be expected to be the cause of flow separation. If this assumption is correct and blading can be obtained which will control the velocity extremes, efficient operation should be attainable for any entrance Mach number.

DETERMINATION OF BLADE SHAPES FOR HIGH MACH NUMBERS

Prescribed velocity distribution. - On the basis of the preceding discussion, the design should control the velocity change that occurs through the shock system. However, the accurate determination of the shock system is questionable, and the control of this item is therefore difficult to achieve. A more convenient approach to the design would be to restrict the over-all velocity change on the suction surface from the peak or maximum velocity to the minimum or exit velocity. The velocity change that occurs through the shock system should then be less than or at most equal to this over-all velocity change providing there is no separation and flow reacceleration.

The selection of the ratio of maximum velocity to exit velocity and the variation of the velocity along the suction surface should ideally be based on considerations of the boundary layers. However, in view of the scarcity of such information, it is convenient to assume the approximate velocity distribution occurring on an airfoil at low speeds which has good performance characteristics. Such a distribution, approximated from the 65-12-10 airfoil at an angle of attack of 8° and a solidity of 1.00 (fig. 56, ref. 7), is shown in figure 8. The ratio of maximum to exit velocity is equal to 1.37. Because of the change in mean velocity occurring across the cascade, the maximum surface velocity is only 1.12 times the entering velocity.

Computation of blade shapes. - The accurate computation of the blade shape for the Mach number range of interest would be extremely difficult. The flow field is a mixed supersonic and subsonic field, and the condition of continuity and the usual assumption of a constant axial velocity across the blade row require that the passage or stream-tube height decrease through the blade row. By applying approximate methods, however, a qualitative picture of the effect of Mach number on blade camber and thickness distribution may be obtained.

The procedure follows that used in channel-flow solutions (ref. 8) with the mean channel velocity assumed equal to the average of the blade surface velocities. From the prescribed velocities on the suction and pressure surfaces,

$$\overline{V}' = (V_s' + V_p')/2 \tag{3}$$

The continuity equation for any point along the channel is

$$\rho_1 V_1' \cos \beta_1' = \overline{\rho V}' \cos \overline{\beta}' \ (1 - nt/2\pi R)H \tag{4}$$

If isentopic flow is assumed, the density may be obtained from the energy equation,

$$\overline{\rho} = \left\{1 + \left(\frac{\gamma-1}{2}\right)\left[M_T^2R^2 - \overline{V}'^2 - 2M_TRV_{\theta,1}\right]\right\}^{\frac{1}{\gamma-1}} \tag{5}$$

The ratio of the stream-tube height H at the exit to that at the entrance is obtained from the continuity equation by using the prescribed conditions of velocity and flow angle at the blade entrance and exit $(nt/2\pi R = 0)$. The distribution of H through the channel may then be specified.

The average flow direction at any point in the channel is obtained by equating the torque due to the pressure difference on the blade surfaces to the change in moment of momentum, or

$$- \sigma_a \int_0^z \Delta pHR \, dz = \gamma \ \rho_1 V_1' \cos \beta_1' \ (R_2V_{\theta,2} - R_1V_{\theta,1}) \tag{6}$$

where the solidity σ_a is defined as the ratio of the axial-blade length to the blade spacing. Equation (6) is solved first for the required solidity by using the prescribed entrance and exit flow conditions, the specified H values, and the Δp obtained from the prescribed velocity distribution. The absolute tangential component V_θ is then obtained for any z position. The average relative flow direction is

$$\overline{\beta}' = \arcsin\left(\frac{M_TR - \overline{V}_\theta}{\overline{V}'}\right) \tag{7}$$

The continuity equation (4) may then be solved for the blade thickness term $nt/2\pi R$ for any z position. If the mean line of the blade is assumed to follow the mean flow path, the blade shape may be obtained.

The assumptions used in the solutions for the required channel flow path have been found to be reasonably accurate for high-solidity blade rows; but for solidities below about 1.5, the results must be considered as qualitative. In any case, the solutions near the leading edge are questionable. Because the method incorporates only the continuity equation, mixed-flow regions may be considered; consequently, this approach was used to furnish general trends that might be expected from high-speed blading.

Example. - The method just described is used to compute the approximate blade shape for a given prescribed velocity distribution for several relative entrance Mach numbers. For the example, the prescribed velocity distribution of figure 8 is used with inlet flow angles of 55°, exit angles of 40°, and a constant axial velocity across the blade row. Blade shapes are determined for incompressible flow and for inlet relative Mach numbers of 0.8, 1.2, and 1.4. Axial absolute inlet flow was assumed ahead of the blade row. The isentropic pressure ratios that would result for relative Mach numbers of 0.8, 1.2, and 1.4 are 1.318, 1.666, and 1.862, respectively.

Because of the density change across the blade row at the higher Mach numbers, the required stream-tube height at the blade exit becomes appreciably smaller than that at the inlet. For a Mach number of 1.4, the ratio of the exit to inlet stream-tube height H is 0.67. A linear variation in H across the blade row was assumed for all cases.

The blade shapes obtained for this example are shown in figure 9. For incompressible flow, the blade resembles the conventional subsonic airfoil with the exception that the leading-edge radius is small. Because of the assumption of similar mean velocity and loading distribution, the mean lines for all Mach numbers are very similar and can be approximated with a circular arc. The principal effect of the increased Mach number is to shift the maximum thickness toward the rear of the blade so that, at a relative entrance Mach number of 1.2, the thickness distribution is nearly symmetrical. As a result, the blade shape obtained for a Mach number of 1.2 can be closely approximated by the double-circular-arc type airfoil. The maximum blade thickness of 8.6 percent of the chord for the incompressible example is reduced to 5.3 percent at a Mach number of 1.4.

The solidity σ necessary to maintain the prescribed velocity ratios ranged from 1.04 for the incompressible case to 1.08 for a Mach number of 1.4.

EXPERIMENTAL RESULTS

Compressor blade shapes similar to those obtained in the example were experimentally investigated over a range of inlet Mach numbers as single-stage rotors. Blade shapes very close to the double circular arc were used in a rotor having a design tip speed of 1000 feet per second and a tip relative Mach number of 1.1. The performance of this rotor is shown in figure 10. The rotor gave good performance at design speed and during part-speed operation.

A blade shape similar to that obtained for an inlet relative Mach number of 1.4 was used in a compressor having a design tip speed of 1400 feet per second. The performance of this rotor is shown in figure 11. Here again, acceptable efficiencies were obtained at the higher tip speeds with no significant sacrifice in part-speed performance. The low values of mass flow per unit frontal area are due to the high inlet hub-tip ratio of 0.7. The discharge conditions at maximum efficiency and design speed gave a Mach number of less than 0.85 at the stators and flow angles of about 45°. The exit distribution was similar to that obtained in the usual transonic rotors. Although the stator Mach number is slightly higher than generally encountered, stator design was not considered a serious problem, since the stator diffusion factor would not be critical at any radius.

The effect of Mach number on performance is illustrated in figure 12 where the efficiency is plotted as a function of tip relative Mach number for two high Mach number compressor rotors. On the basis of the limited data available, the effect of Mach number on efficiency appears to be small when the blade shapes and solidities are chosen to limit the maximum velocity ratio (or blade loading) on the blade surfaces. In both cases, the drop in efficiency at design speed was associated with a rapid rise in the diffusion factor near the rotor tip.

CONCLUSIONS

In order to realize the desirable compressor characteristics of high flow capacity and high stage pressure ratio, Mach numbers relative to the rotor blading above 1.0 must be utilized. By operating in the high Mach number range, the need for guide vanes to impart a large prerotation into the air is eliminated, and high-pressure-ratio stages can be designed without excessive tip loading. The allowable losses arising from the use of supersonic relative velocities can increase in the same proportion as the permissable work input without the efficiency level being reduced.

On the basis of limited experimental observations, the control of the blade-surface velocity extremes appears to be the critical factor in the design of high-speed compressor blading. A qualitative picture of the effect of Mach number on blade shapes having a prescribed velocity diagram which controlled the velocity peaks was obtained by the use of a very approximate design method, and the following observations were made:

1. The stream-tube height across the blade row must be appreciably reduced for high Mach numbers.

2. The principal effects of Mach number were to displace the position of maximum thickness toward the rear of the blade and to reduce maximum blade thickness. The blade leading-edge radius was small for all examples.

3. For a relative entrance Mach number of 1.2, the blade-thickness distribution was nearly symmetrical and the blade shape could be closely approximated by use of a double-circular-arc airfoil.

4. The required blade solidity remained approximately constant as the Mach number was varied.

Limited experimental results showed rotor efficiencies of above 90 percent for rotor tip Mach numbers up to 1.4, and no sacrifice in low-speed performance was observed.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, December 8, 1953

APPENDIX - SYMBOLS

The following symbols are used in this report:

D diffusion factor (see eq. (2))

H ratio of stream tube height at any point to that at blade entrance

M_T rotor tip speed made dimensionless by upstream stagnation sonic velocity

n number of blades

P total pressure made dimensionless by upstream stagnation pressure

p static pressure made dimensionless by upstream stagnation pressure

Δp dimensionless pressure difference between pressure and suction surface

R radius made dimensionless by tip radius

t blade thickness made dimensionless by rotor tip radius

V absolute velocity made dimensionless by upstream stagnation sonic velocity

V' relative velocity made dimensionless by upstream stagnation sonic velocity

ΔV_θ change in absolute tangential velocity across the blade row made dimensionless by upstream stagnation sonic velocity.

z axial position made dimensionless by axial length of blade.

β absolute flow angle measured from axis of rotation

β' relative flow angle measured from axis of rotation

γ ratio of specific heats

η_{ad} adiabatic efficiency

ρ density made dimensionless by upstream stagnation density

σ blade solidity defined as ratio of blade chord to blade spacing

σ_a blade solidity defined as ratio of axial blade length to blade spacing

Subscripts:

s suction surface

p pressure surface

0 upstream stagnation

1 ahead of blade row

2 behind blade row

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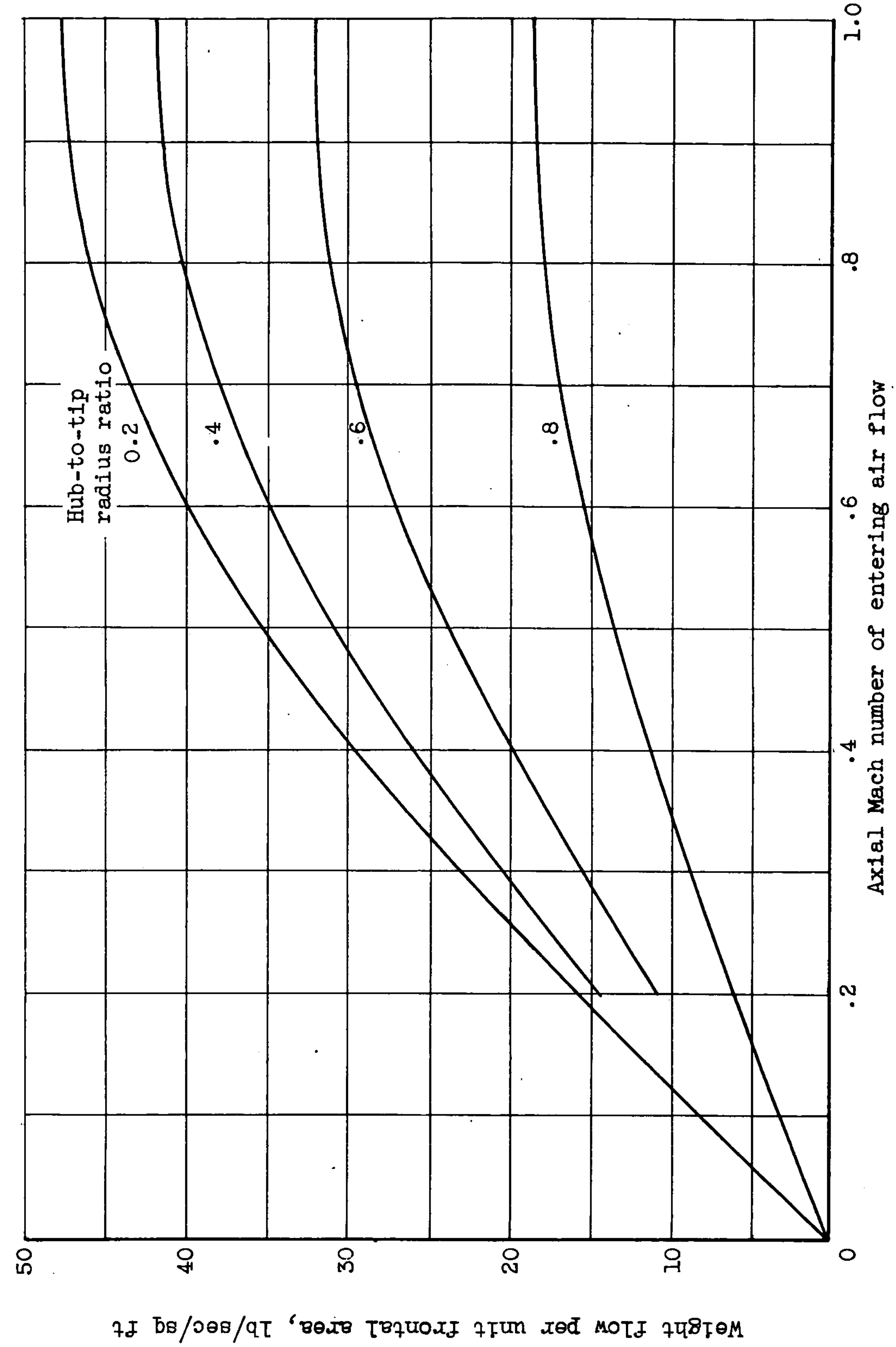


Figure 1. - Air-flow capacity of axial-flow compressors.

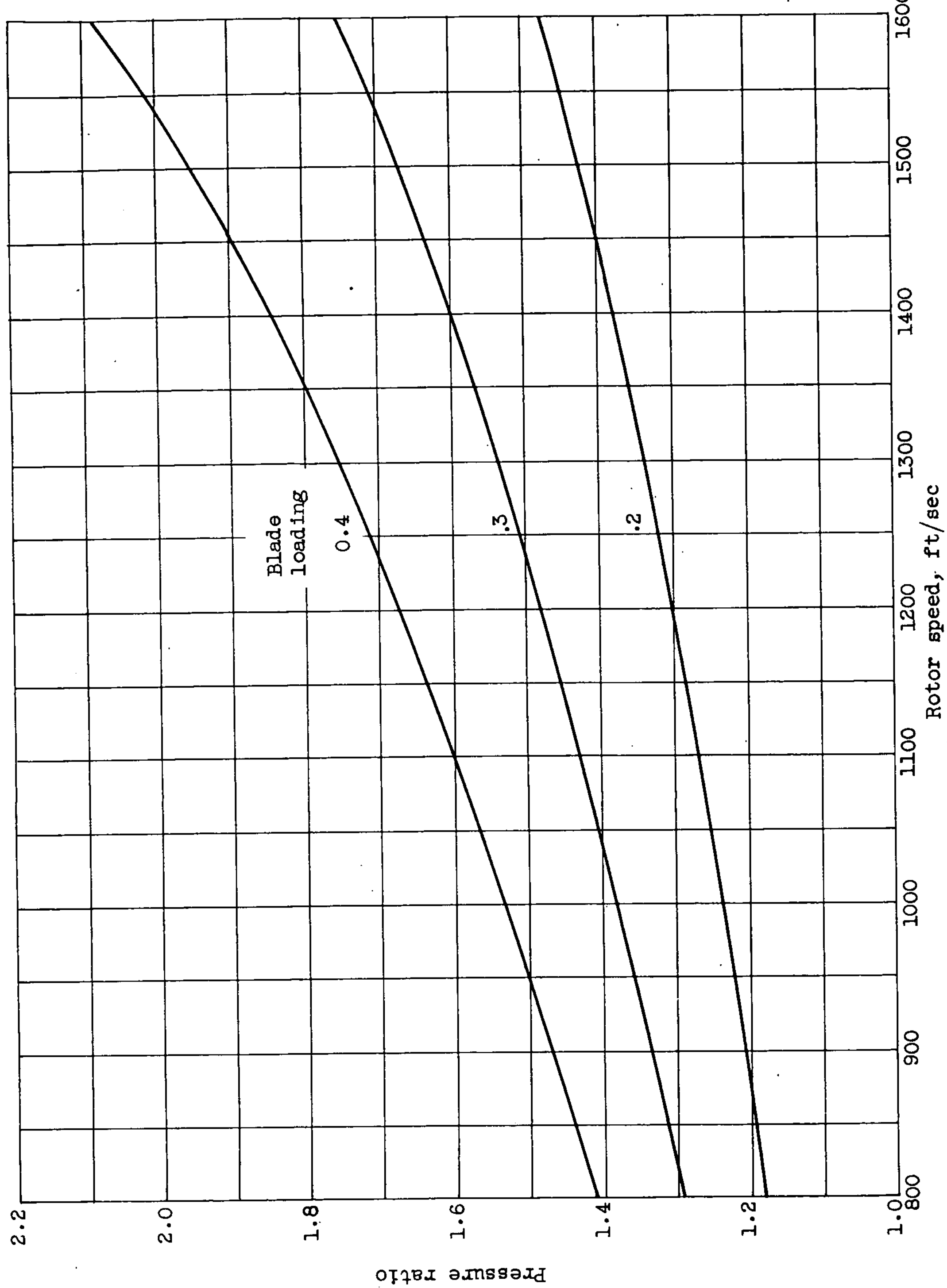


Figure 2. - Variation of pressure ratio of axial-flow compressors with rotor speed and blade loading.

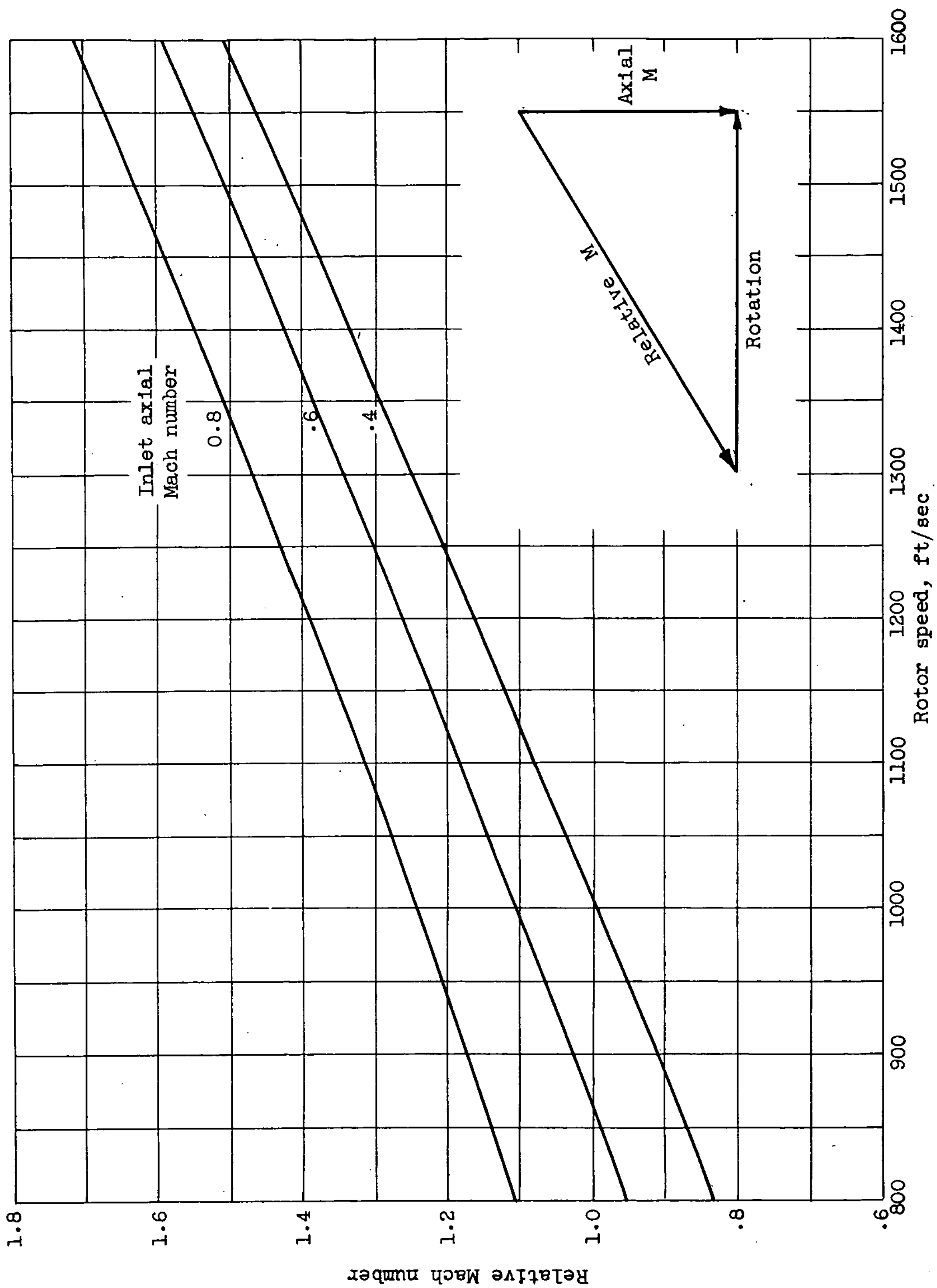


Figure 3. - Relative inlet Mach number characteristics of compressors.

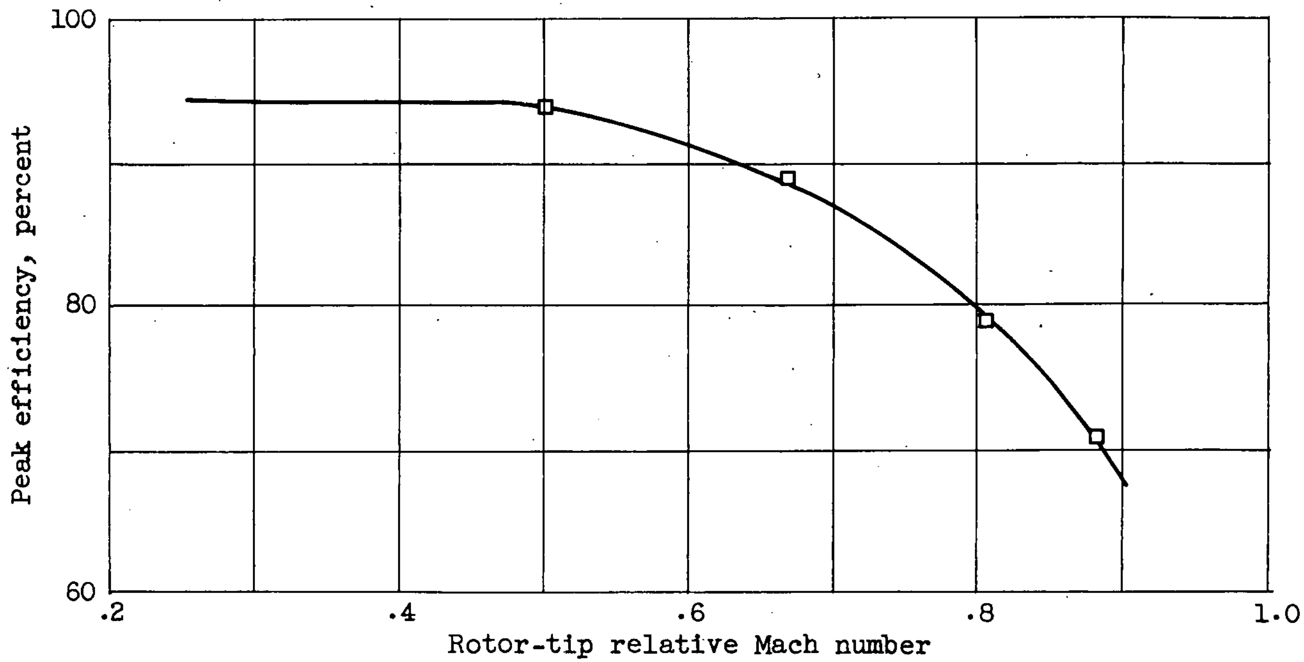


Figure 4. - Variation of efficiency with Mach number for symmetrical-velocity-diagram design. (Data from ref. 2.)

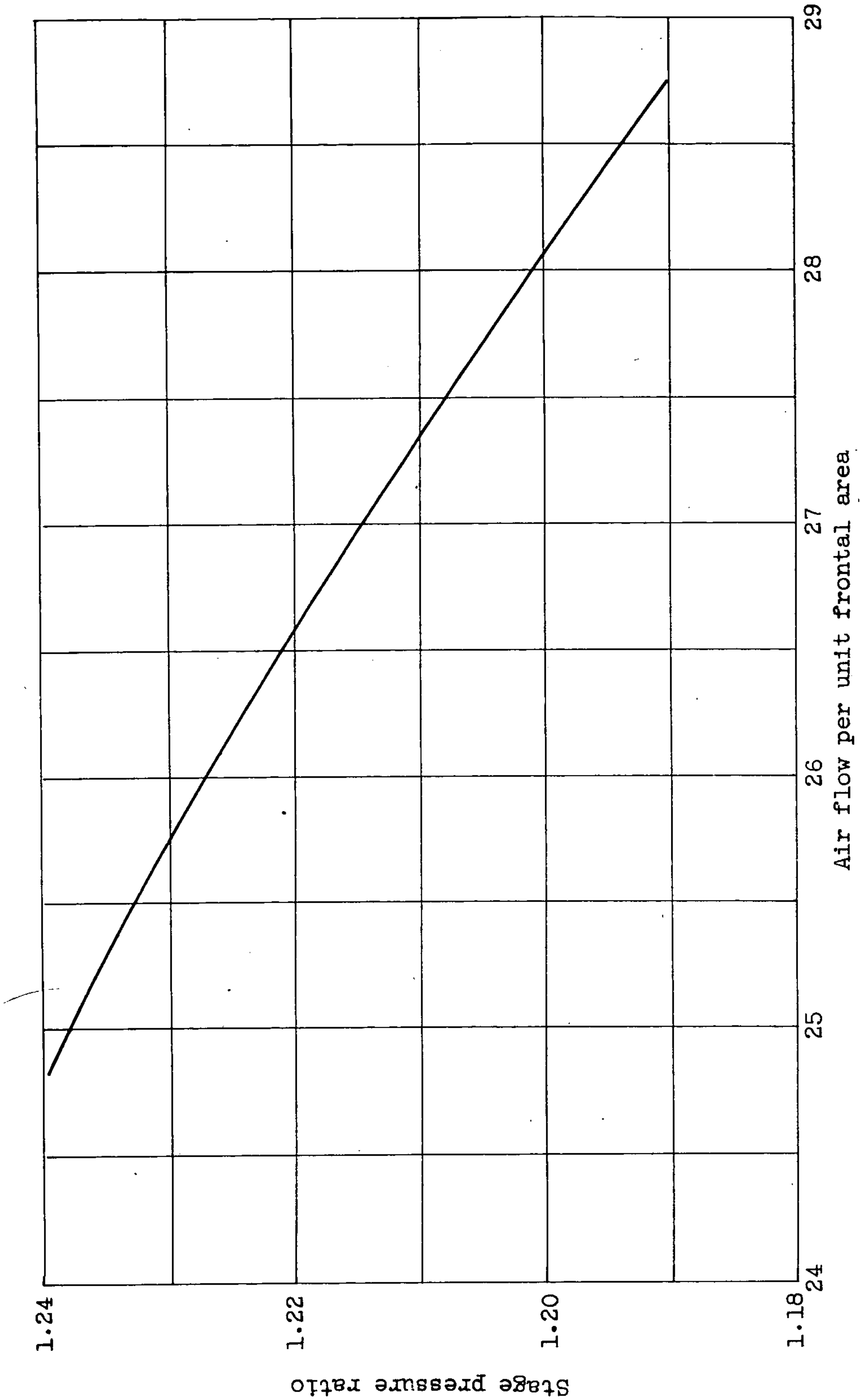
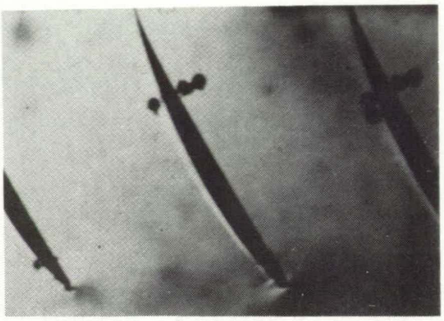
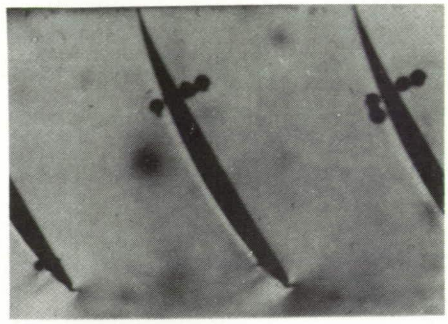


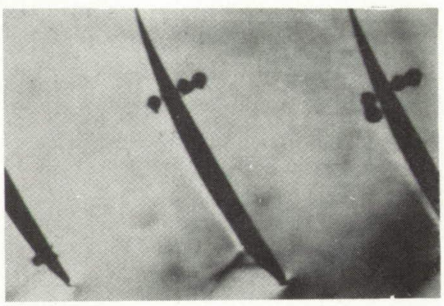
Figure 5. - Design compromise for conventional subsonic compressor. Hub-tip radius ratio, 0.5.



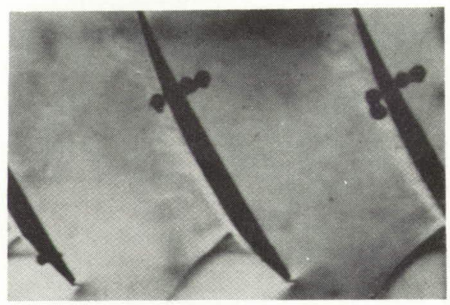
Mach number, 0.70



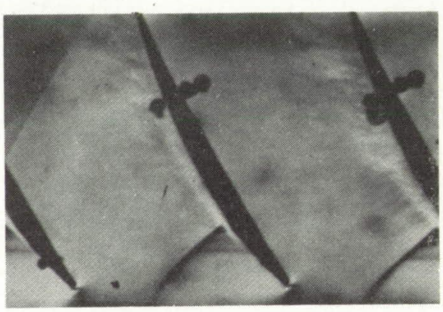
Mach number, 0.75



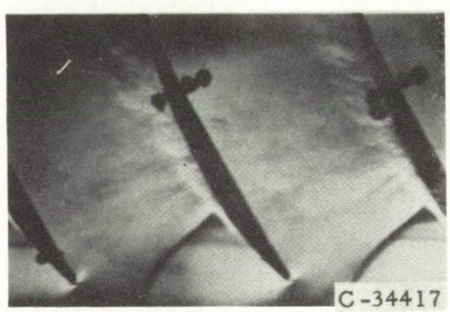
Mach number, 0.82



Mach number, 0.88



Mach number, 0.93



Mach number, 0.95

Figure 6. - Schlieren photographs for a range of Mach number. Cascade of NACA 65-806 blower blades. Angle of attack, 16.5°; absolute flow angle, 45°; blade solidity, 1.5.

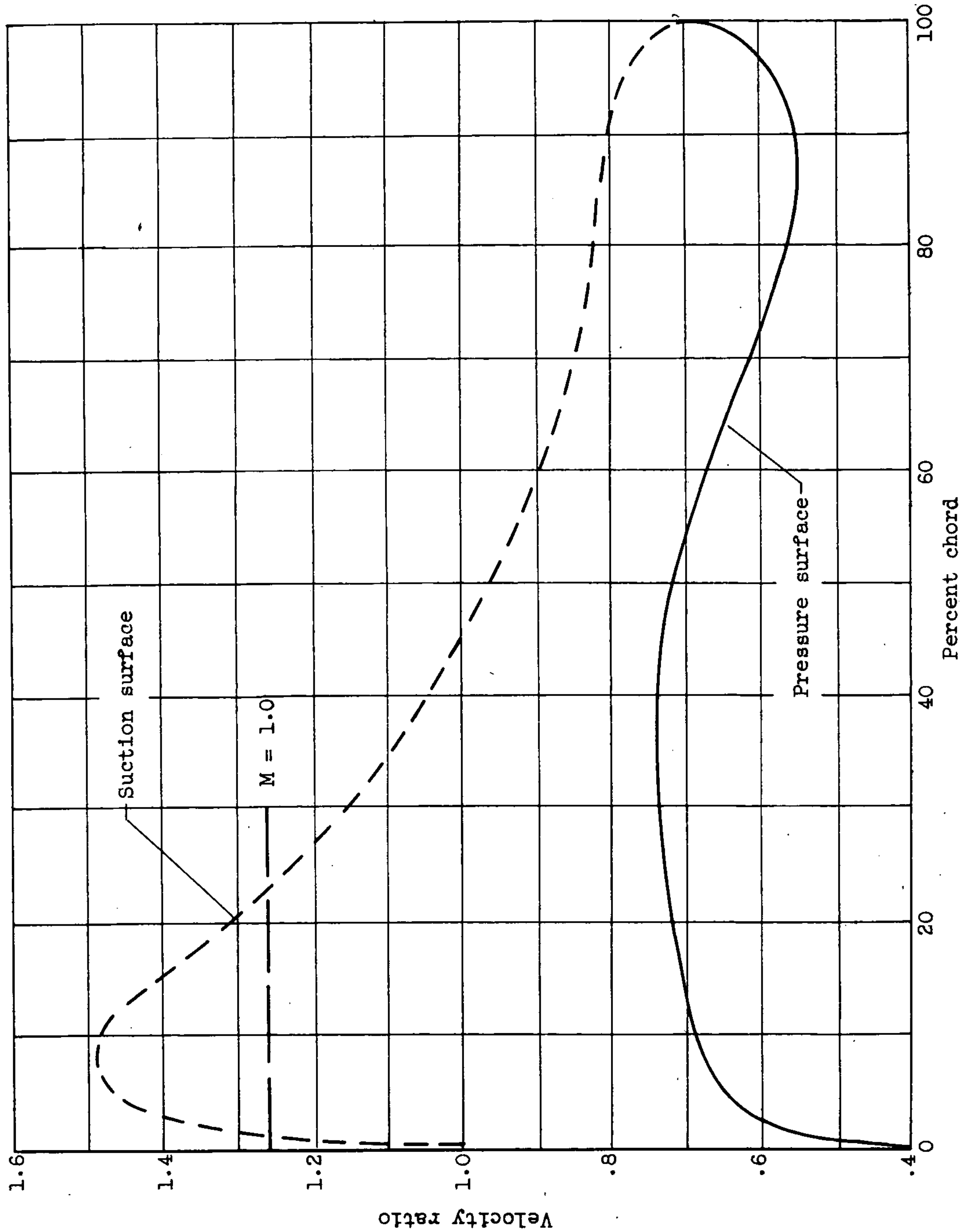


Figure 7. - Velocity distribution over subsonic airfoil at high Mach number.

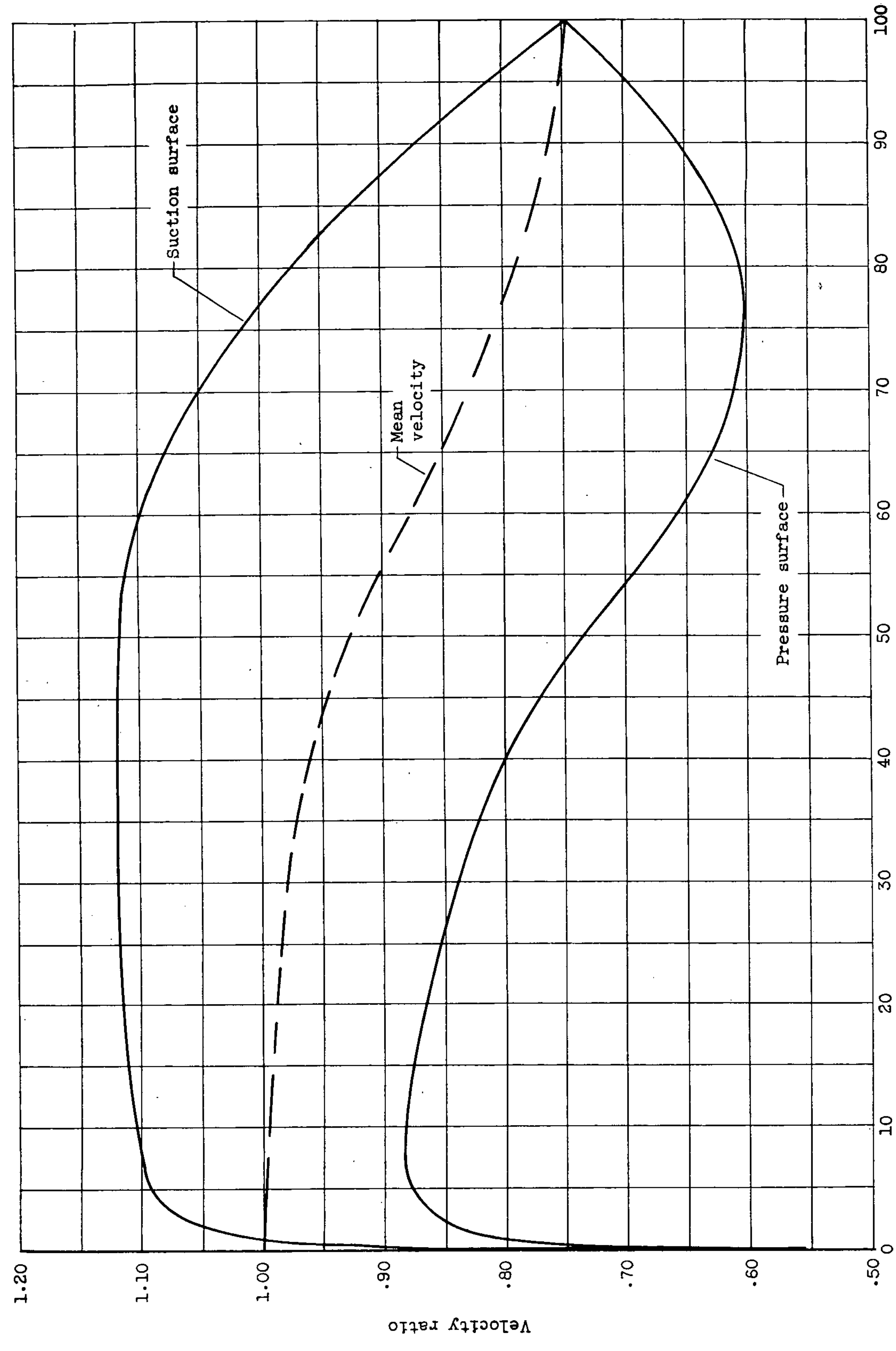


Figure 8. - Prescribed velocity distribution over airfoil in cascade.

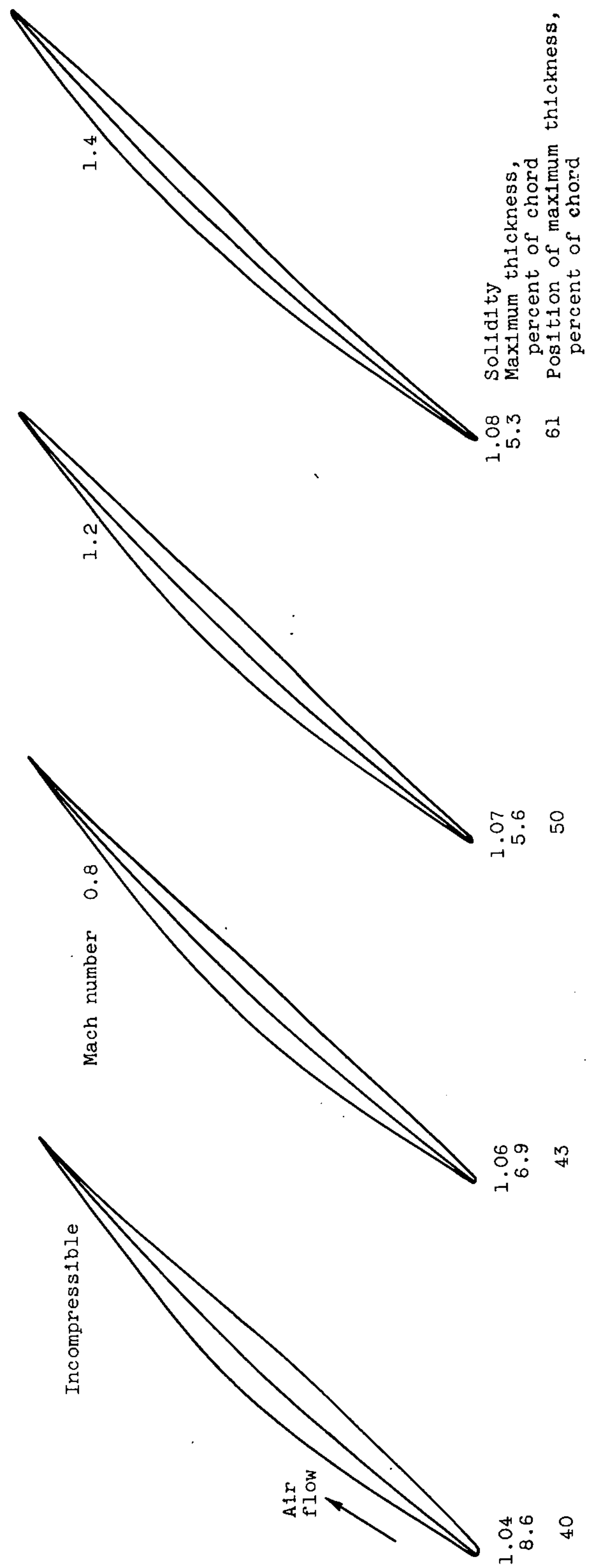


Figure 9. - Computed blade shapes for prescribed velocity distribution.

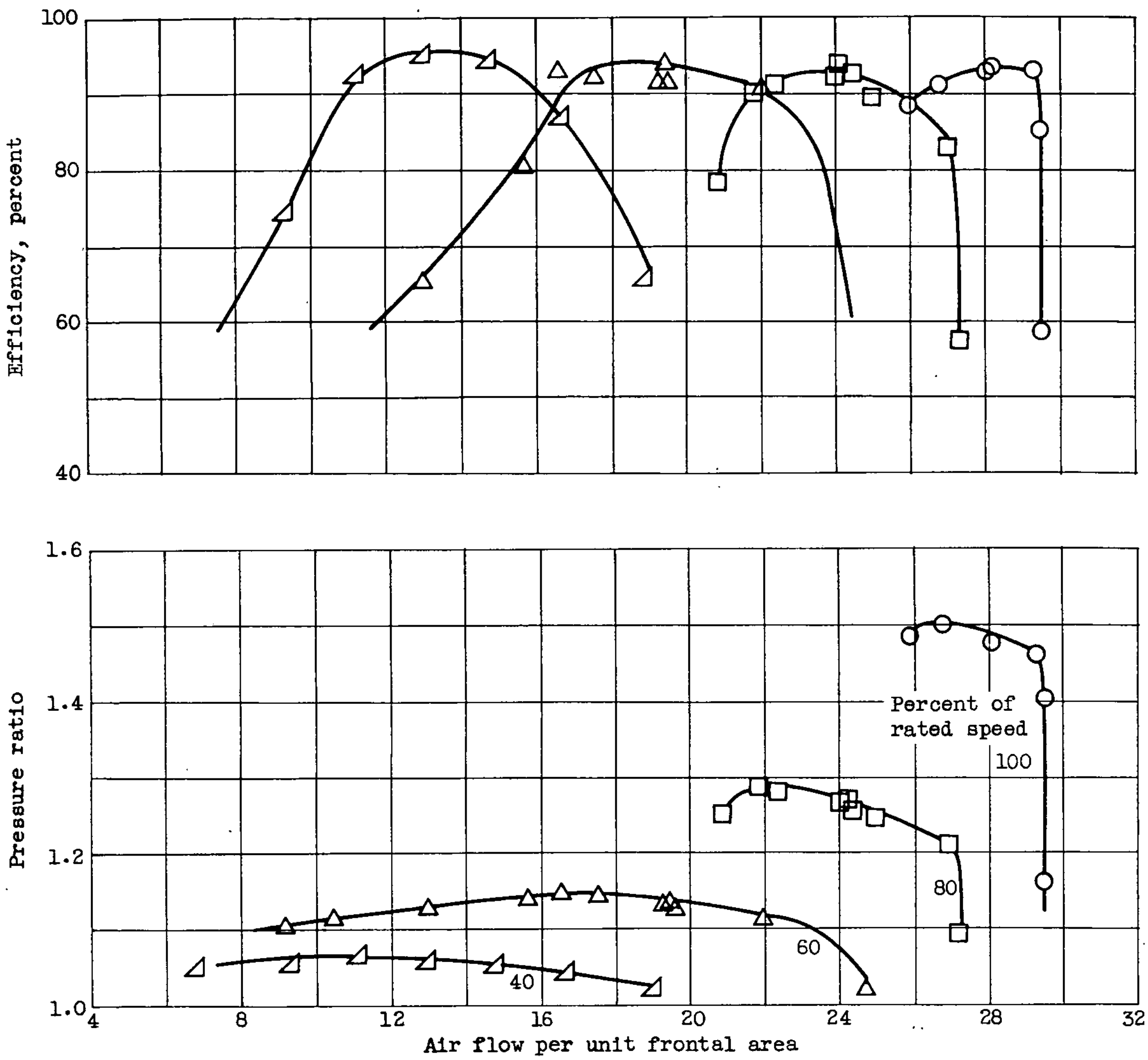


Figure 10. - Performance of transonic rotor designed for tip speed of 1000 feet per second. Hub-tip radius ratio, 0.525.

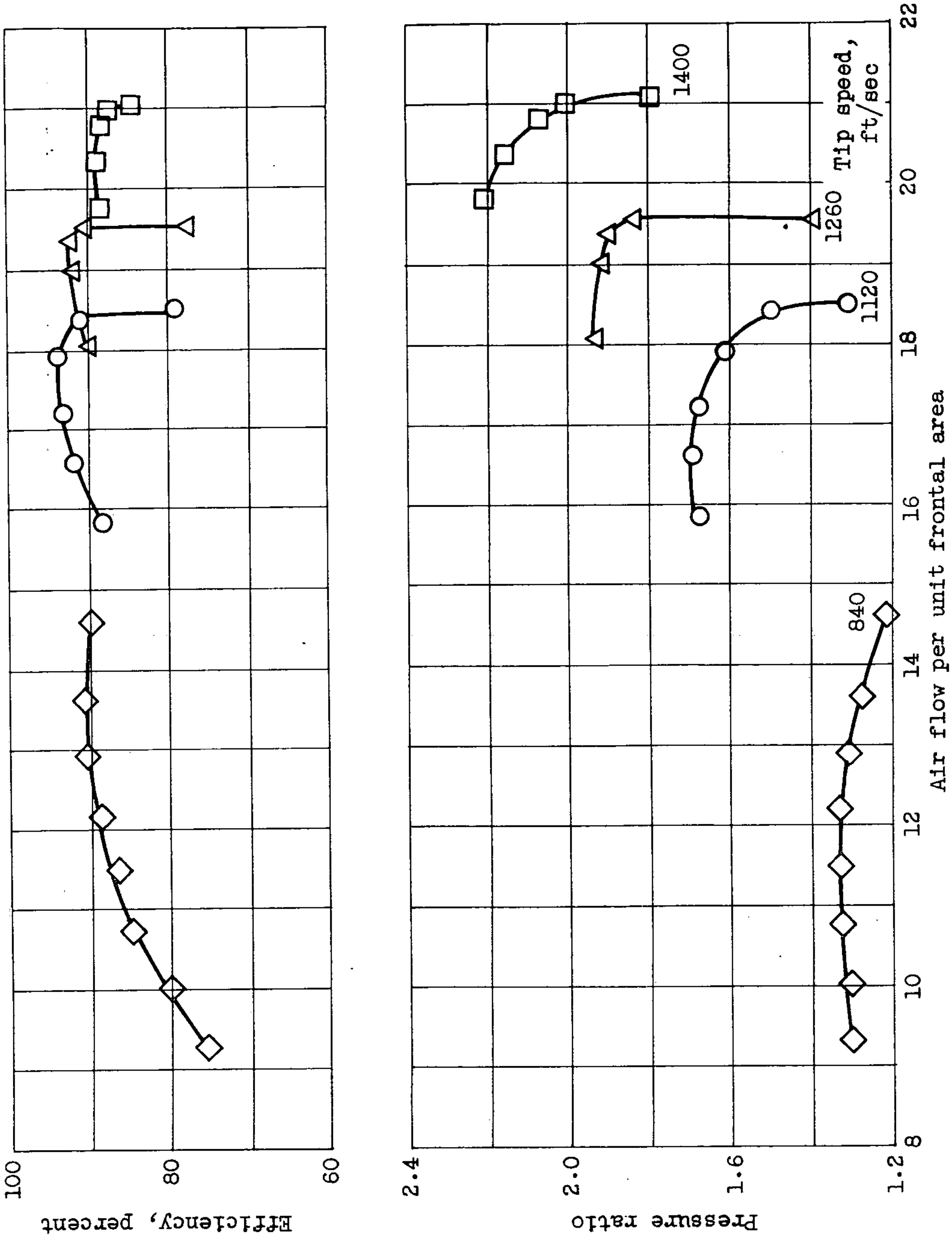


Figure 11. - Performance of compressor rotor designed for tip speed of 1400 feet per second. Hub-tip radius ratio, 0.7.

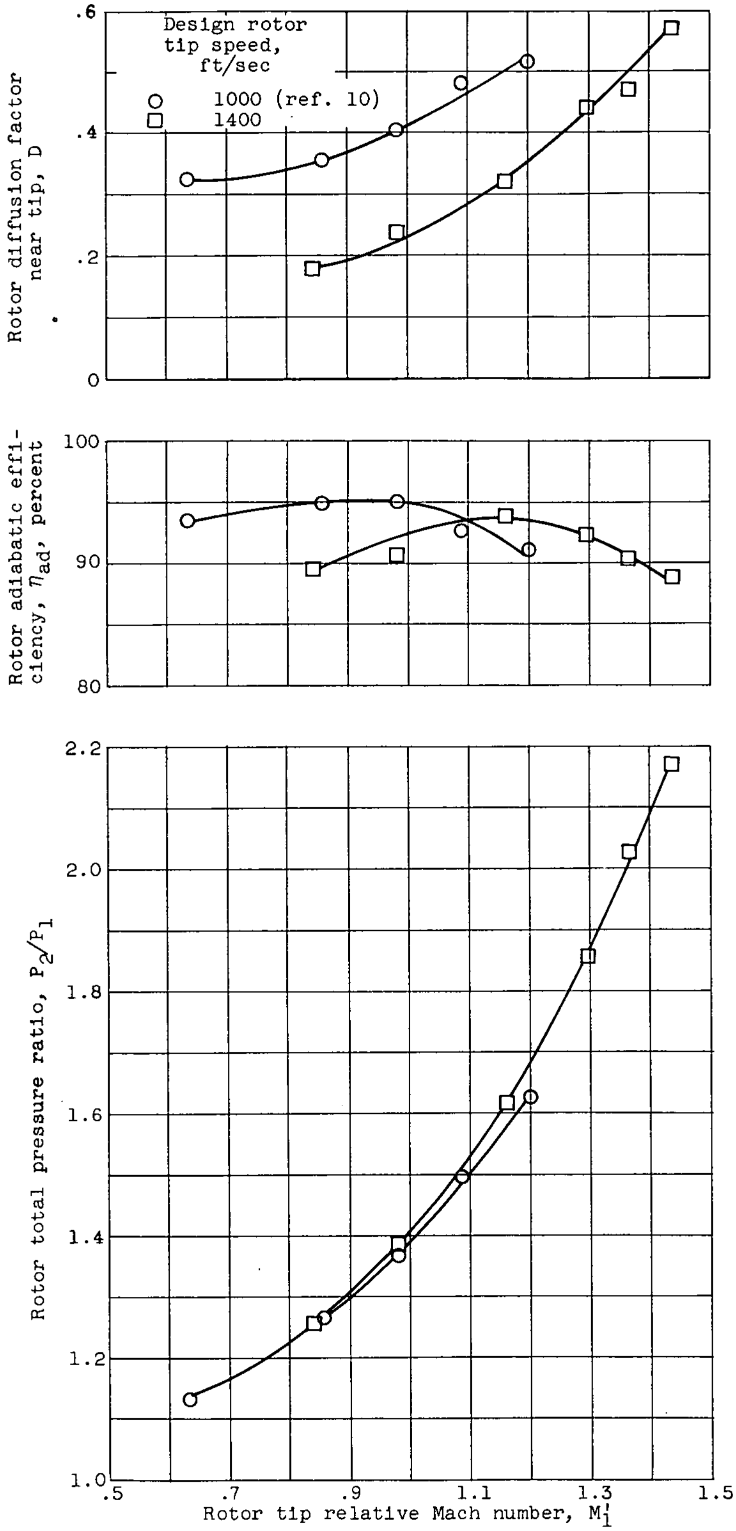


Figure 12. - Experimental performance of two compressor rotors having high Mach numbers.